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Estimation of Aerodynamic Load Distributions on the F/A-18 Aircraft Using a CFD Panel Code

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ABSTRACT

The computational fluid dynamics panel method code VSAERO has been used to estimate the aerodynamic load distributions on the stabilators, fins and aft fuselage of the F/A-18 aircraft for steady sideslip and steady roll conditions. Three separate VSAERO models have been developed to obtain results for these conditions. These are a symmetric F/A-18 model, a starboard fin model and a starboard stabilator and fin model. Difficulties in modelling the vortices separating from the leading edges of the stabilators and fins at moderate angles of attack have restricted the range of conditions for which the method has been used.

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Executive Summary

The International Follow On Structural Test Programme (IFOSTP) is a combined programme by Canada and Australia to assess the fatigue life of the F/A-18 aircraft under service conditions. Australia's contribution involves conducting a fatigue test of the aft fuselage, fins and stabilators. The loading to be applied in these tests is intended to be representative of the actual service usage of the aircraft by the Canadian Forces and the Royal Australian Air Force.

Information on aircraft loading is being obtained from inflight strain gauge measurements, while detailed load distribution information is to be provided by computational and wind-tunnel methods. Air Operations Division was tasked with supplying the aerodynamic load distribution information for the aft fuselage, fins and stabilators of the F/A-18 aircraft. This information was required to cover the operational flight envelope of the F/A-18. Since little data were available from the aircraft manufacturer for this purpose, the approach adopted was to utilise the available resources of DSTO's low speed and transonic wind tunnels and also the computational fluid dynamics (CFD) program, VSAERO. The results obtained from the VSAERO program are presented in this report.

The VSAERO program is a mathematical code based on a surface singularity panel method which models inviscid incompressible flow. Complex geometrical configurations, such as the F/A-18, are relatively easy to input to VSAERO. The program outputs the aerodynamic pressure distribution over the entire aircraft surface, as well as non-dimensionalised total loads. When VSAERO is applied within strict limits these results will accurately represent the full size aircraft's aerodynamic pressure distribution, and also the non-dimensionalised total loads. When developing any VSAERO model, part of the work involved is to establish limits within which the VSAERO model gives realistic surface pressure distributions and non-dimensionalised total loads for each particular aircraft configuration.

The aim of the task was to use the VSAERO program to provide preliminary estimates of the pressure distributions around the aft fuselage, stabilator and fins, and to produce estimates for cases that the wind tunnels could not cover. As the wind tunnel tests were not designed to produce data for rolling manoeuvres, or to predict the effects of rudder deflection, two specialised geometry models were developed for these particular cases from the original F/A-18 symmetric model. Hence three individual models were used for this task, these being the Symmetric F/A-18 model, the F/A-18 Starboard Stabilator and Fin model and the F/A-18 Starboard Fin model.

While establishing the limits for these three models, discrepancies were observed between the VSAERO predictions and wind tunnel results for moderate angles of attack. Swept planform surfaces, with sharp leading edges, are known to promote flow separation and subsequent vortical flow. Both the fin and stabilator have swept planforms, with sharp leading edges, which cause vortical flow over these surfaces. At high angles of attack, or large control surface deflections, the vortical flow has been difficult for the VSAERO program to model in a realistic manner. Hence the VSAERO models have been applied within a reduced envelope for valid computations.

Overall, the VSAERO predictions provided valuable initial information for establishing wind-tunnel test requirements and data handling and data presentation formats for the F/A-18 aerodynamic pressure distribution task.

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Howard completed his Bachelor of Engineering in Aeronautical Engineering at RMIT in 1988, and joined the Aeronautical Research Laboratory (Salisbury) in 1989. He worked on aerial towed targets before being seconded to Air Operations Division in 1990 to work on F/A-18 aerodynamic load distributions for IFOSTP. He has accumulated experience in wind tunnel testing as well as in the computational fluid dynamics techniques used in this study.

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Notation

b	wing semispan (= 5.70 m)
b_{fin}	fin span (= 2.41 m)
b_{stab}	stabilator semispan (= 2.24 m)
B_{jk}	matrix of influence coefficients determined by VSAERO
BM_{fin}	fin bending moment (Nm)
$\bar{c}$	wing mean aerodynamic chord (= 3.51 m)
$\bar{c}_{fin}$	fin mean aerodynamic chord (= 2.13 m)
$\bar{c}_{stab}$	stabilator mean aerodynamic chord (= 1.91 m)
c	panel chord on fin or stabilator surface (m)
C_{lf}	fin local section lift coefficient, $\sum_{te}^{le} (C_p c)/\bar{c}_{fin}$
C_{ls}	stabilator local section lift coefficient, $\sum_{te}^{le} (C_p c)/\bar{c}_{stab}$
C_{jk}	matrix of influence coefficients determined by VSAERO
C_{jm}	matrix of influence coefficients determined by VSAERO
C_{BMf}	fin bending moment coefficient $(BM_{fin}/qS_{fin}\bar{c}_{fin})$
C_D	total aircraft drag coefficient (D/qS_{wing})
C_L	total aircraft lift coefficient (L/qS_{wing})
C_{Lf}	fin normal force coefficient (L_{fin}/qS_{fin})
C_M	total aircraft pitching moment coefficient $(M_y/qS_{wing}b)$
C_{Mf}	fin pitching moment coefficient $(M_{yf}/qS_{fin}b_{fin})$
C_p	pressure coefficient $((p - p_o)/q)$
D	drag force (N)
L	lift force (N)
L_{fin}	fin normal force (N)
M	Mach number
M_y	pitching moment (Nm)
M_{yf}	fin pitching moment (Nm)
n	unit normal to surface
P	point in the volume analysed by VSAERO
p	static pressure at point in flow (Pa)
p_o	static pressure of undisturbed flow (Pa)
$p_{x'}$	roll rate about the VSAERO x′-axis (rad/s)
q	dynamic pressure, $1/2\rho V_\infty^2$ (Pa)
r	radius from the point P to the surface normal point (m)
S	surface boundary of the volume (VSAERO)
S_{wing}	wing reference area (= 37.16 m^2)
S_{fin}	fin reference area (= 4.84 m^2)
S_{stab}	stabilator reference area (= 8.18 m^2)
V_∞	freestream velocity (+ve in x direction for $\alpha = 0°$, $\beta = 0°$, m/s)
x, y, z	VSAERO station coordinates (in)
x', y', z'	VSAERO station coordinates used in Starboard Stabilator and Fin Model (in)
Y_H	local y coordinate on stabilator (m)
Z_V	local z coordinate parallel to fin surface (m)
α	angle of attack (degrees)
β	angle of sideslip (degrees)
δ_r	rudder deflection (+ve trailing edge to starboard, degrees)
ϕ	velocity potential calculated in VSAERO
$\Omega_{x'}$	normalized rotation about VSAERO x′-axis, $p_{x'}b/2V_\infty$
ρ	density of air (kg m^{-3})

Abbreviations

le	leading edge
te	trailing edge
thick	thick surface as defined by VSAERO
thin	thin surface as defined by VSAERO
LEX	Leading Edge Extension
$NPAN$	number of surface panels defining a VSAERO model
$NWAKE$	number of wake panels defining a VSAERO model wake

Axes Systems

- The positive sign convention for Roll Rotation on the Starboard Stabilator and Fin Model is presented in figure 3.
- The sign conventions for the forces and moments acting on the stabilator and fin are presented in figure 5.
- The reference stations, indicating where pressure distributions have been compared in this report, are presented in figure 6.

Reference Points for Forces and Moments

Reference Point Description	X Coordinate inches	Y Coordinate inches	Z Coordinate inches
Aircraft Total Forces and Moments	457.061	0.0	107.725
Fin Forces and Moments	552.16	39.75	118.38
Stabilator Forces and Moments	651.60	41.92	99.32

Note: Aircraft dimensions provided by the manufacturer are given in inches and have not been converted to SI units.

1 Introduction

The International Follow On Structural Test Programme (IFOSTP) is a combined effort by Canada and Australia to assess the fatigue life of the F/A-18 aircraft under service conditions. Australia's contribution involves conducting the fatigue test of the aft fuselage and empennage. The loading to be applied in these tests is intended to be representative of the actual service usage of the aircraft by the Canadian Forces and the Royal Australian Air Force.

Information on aircraft loading is being obtained from inflight strain gauge measurements, while detailed load distribution information is to be provided by computational and wind-tunnel methods. Air Operations Division (AOD) was tasked with supplying the aerodynamic load distribution information for the aft fuselage and empennage of the F/A-18 aircraft. This information was required to cover the operational flight envelope of the F/A-18. Since little data were available from the aircraft manufacturer for this purpose the approach adopted was to utilize the available resources of DSTO's low speed and high speed wind tunnels and also the computational fluid dynamics (CFD) code VSAERO[1]. The results obtained from VSAERO are dealt with in this report.

VSAERO is a panel method code which models inviscid incompressible flow. The relative ease of application of panel codes to complex configurations, and their computational speed compared to higher order codes are the main attractions of these codes. The disadvantages associated with them are that complex flows due to compressibility or viscous effects are not represented directly, and this can limit their range of application.

The aim of the task was to use the VSAERO F/A-18 models to provide preliminary estimates of the pressure distributions around the aft fuselage, stabilator and fins, and to produce estimates for cases that the wind tunnels could not cover. Specifically, the wind tunnel tests were not designed to produce data for rolling manoeuvres or to predict the effects of rudder deflection.

[1] Analytical Methods Inc., Washington, U.S.A.

2 A Brief Description of VSAERO

The VSAERO program is based on a surface singularity panel method. This program models inviscid incompressible flow about arbitrary configurations and includes options for modelling the nonlinear effects of wake roll-up and compressibility. The method solves the boundary integral equation obtained from the application of Green's Theorem to Laplace's equation. Normal velocities on the surfaces bounding the flow must be specified (i.e. Neumann boundary condition). The equation is given by

$$4\pi\phi(P) - \int_{S_{thick}} \phi \frac{\delta}{\delta n}\left(\frac{1}{r}\right) dS - \int_{S_{thin}} \Delta\phi \frac{\delta}{\delta n}\left(\frac{1}{r}\right) dS = -\int_{S_{thick}} \frac{1}{r}\frac{\delta\phi}{\delta n} dS.$$

Here ϕ represents the velocity potential, P represents some point in the volume, S is the surface boundary of the volume (model), n is the surface normal, and r is the radius from the point P to the surface normal point. This equation is discretised to a number of quadrilateral panels covering the body surface. These panels have the doublet and source singularities distributed in a piecewise constant form. The value of the source on each panel forms the unknown in the equation. Hence the number of panels used to define a surface is also the number of unknowns in the equation.

In its discretised form the equation is given by

$$\sum_{k=1}^{NPAN} (\phi_k C_{jk}) + \sum_{m=1}^{NWAKE} (\Delta\phi_{w_m} C_{jm}) = \sum_{k=1}^{NPAN} \frac{\delta\phi}{\delta n_k} B_{jk}; j = 1, NPAN.$$

$NPAN$ and $NWAKE$ are the number of surface and wake panels, respectively. C_{jk} and B_{jk} are the influence coefficients for the constant source and doublet distributions, respectively. C_{jm} represent the influence coefficients from the wake. The equation is further simplified by eliminating the wake strengths as unknowns. This is achieved by accounting for the wake strength in terms of the potential values on the body shedding panels, and by specifying the variation of the wake strength along each wake column. Further details about VSAERO are available in Reference [1].

This report contains two other VSAERO terms which are defined here for convenience. The terms are patches and components. A patch is a set of panels defining a section of the surface geometry, usually where the surface has continuous slope. Where the geometry has discontinuities in the slope, a number of patches may be required to define the surface. A component is defined by a group of patches and may represent a whole wing, or fuselage. The main features of a component are that it may be translated and rotated within the global axis system, and that a summary of the forces and moments acting on the component is given in the output.

All results presented in this report were obtained using VSAERO version E.2.

3 Description of the VSAERO Models

3.1 Symmetric F/A-18 Model

The F/A-18 VSAERO model has been under development at AOD since 1989. The original model consisted of the wing and fuselage represented by 1492 panels. References [2] and [3] give details of this model.

Figure 1 shows the symmetric F/A-18 VSAERO model that was used for this task. The symmetric F/A-18 model has only the starboard side geometry defined (i.e. half model) using 2957 panels, and a reflection plane is imposed at the aircraft centreline during computation. The term 'symmetric' refers to the aerodynamic cases that may be computed for realistic results using this particular model. Asymmetric aerodynamic cases involving rolling and sideslip could not be covered with this model and therefore separate models were developed for these particular flight conditions. Rudder deflection cases have been computed using this model and the results are presented in section 4.3.

The original F/A-18 VSAERO model required many modifications to obtain a version suitable for this task. The modifications were performed in a number of stages.

Initial work involved checking the original F/A-18 VSAERO model geometry with the reference stations and sections provided in Reference [4] and also incorporating the modifications for the movable control surfaces (i.e. flaps, aileron, rudder and stabilator). Particular attention was paid to the panelling of the aft fuselage, fin and stabilator. Surface intersections between the fin and fuselage, and also the wing and fuselage were made using a program capable of automatically generating the required intersection data. Even with this program the changes in panelling were time consuming to incorporate in the model. General rules regarding panelling a VSAERO model were observed while making these modifications, such as avoiding extreme changes in panel densities within a given patch. Also excessively skewed panels were avoided, where possible, by aligning patch edges with the fuselage centreline. Panel density on the fin, stabilator and wing surfaces was varied to match the expected variation in flow properties. Hence more panels were required at the leading edges to define the rapid changes in flow properties there. In the case of the fin and the wing, this panel distribution was then carried over to the fuselage. The stabilator is an all-moving surface that does not intersect the fuselage, so matching the panelling from the stabilator to the fuselage was not required.

With the geometry changes completed, the next step was to specify the options required in the basic input. Checks for appropriate panel neighbour relations across patch boundaries, panel normal directions, and also the wake specifications were made, and corrections implemented as required. Options specifying engine intake flow and jet exhaust flow are included in the model. A 'flow-through' condition was specified in which the ratio of intake and exhaust velocity to freestream velocity was set equal to unity. This value was selected as it approximately matched the condition used for the tests in the Low Speed Wind Tunnel reported in Reference [5]. Previous work on the effect of engine intake velocity on the overall lift coefficient (Reference [2]) was available to verify this selection. The original VSAERO model of the F/A-18 was used for the work in Reference [2].

The final stage in development of the model involved obtaining preliminary aerodynamic results, and optimizing the model for minimum computational time. To

minimize computational time the patches were re-ordered in the input file, so that non-zero values in the influence coefficient matrix were located close to the matrix diagonal. Several cases were solved at different angles of attack, and the number of iterations required for the matrix solver to converge was recorded. The input parameter for the matrix solver over-relaxation factor was varied for each case. A significant reduction in the number of iterations required for the matrix solver to converge, and hence a reduced computational time, was achieved by using the optimum value for the over-relaxation factor. Further information on the method is given in Reference [6].

The process of obtaining satisfactory results from the symmetric F/A-18 VSAERO model was a time consuming task, due to the difficulty in defining acceptable wakes. The wakes are flexible to allow alignment with the local flow. This condition also allows the wake to pierce a surface, causing erroneous results for panel pressures. To avoid this problem a wake may be constrained by specifying a fixed wake geometry to a point downstream of the previously affected surface. This method was used to prevent the wake interference between the wing wakes and the stabilator, and between the inboard wing wake and the side of the fuselage on the symmetric F/A-18 VSAERO model.

It should be noted that the final version of the model did not include the Leading Edge Extension (LEX) wake. Attempts had been made, with earlier versions of the symmetric F/A-18 VSAERO model, to include this wake at the LEX leading edge. Generally the LEX wake failed to roll-up in an acceptable manner and pierced the fin and fuselage panels, producing unrealistic results. As the LEX vortex contributes significantly to the lift at higher angles of attack, its omission limited the range of angles of attack that could be covered.

3.2 F/A-18 Starboard Stabilator and Fin Model

Figure 2 shows the VSAERO model of the F/A-18 starboard stabilator and fin used to obtain asymmetric aerodynamic results. This model was developed to provide estimates of load distribution during steady rolling manoeuvres, as the symmetric F/A-18 model was unusable for these asymmetric aerodynamic cases. This model was copied directly from the symmetric F/A-18 model input file, with any panels and wakes not associated with the fin and stabilator deleted. The fin and stabilator were repositioned in the z direction as indicated in figure 3 to correctly model the roll rotation about the aircraft's centre of gravity. The panel density was changed following a brief study of the effect of panel density on spanwise lift distributions. The total number of panels defining the fin was increased from 450 to 486 panels, and similarly for the stabilator from 240 to 375 panels. It was recognized that this model had limitations when compared with the symmetric F/A-18 model as the effects of the fuselage and the downwash from the wing were not represented. Hence all results obtained were considered to be estimates of the load distribution due to a steady rolling manoeuvre.

3.3 F/A-18 Starboard Fin Model

The F/A-18 starboard fin model, shown in figure 4, was used to obtain results for sideslip cases with rudder deflection. This model was developed from a copy of the starboard stabilator and fin model input file with the stabilator panels and wakes removed. The total number of panels defining the fin was increased from 486 to 589 to increase the accuracy of results.

Two assumptions are inherent in the use of a fin-only model. These are that the air flow seen at the fin is assumed to be undisturbed by wakes or vortices from upstream and secondly the effects from the fuselage and wing are ignored. While these assumptions will lead to some inaccuracies, it was anticipated that this model would be capable of providing satisfactory first order estimates of the fin load distributions due to rudder deflections.

4 Results and Discussion

4.1 Symmetric F/A-18 Model

This model was used to calculate lift, drag and pitching moment coefficients as well as surface pressures for $\alpha = -4°, 0°, 4°, 8°$ at Mach 0.2 and Mach 0.6, with all control surfaces undeflected. As the symmetric F/A-18 model is a half geometry model, VSAERO outputs total forces and moments by doubling the values obtained from the half model. VSAERO accounts for the effect of Mach number by applying a compressibility correction to the inviscid solution. All results presented in this report were obtained using the Karman-Tsien formula (Reference [7]) to correct for compressibility effects.

The sign convention for the forces and moments acting on the stabilator and fin are marked on figure 5. The reference stations, indicating where pressure distributions from VSAERO and wind tunnel results (Reference [5]) have been compared in this report, are presented in figure 6.

Figure 7 presents VSAERO results for aircraft total lift, drag and pitching moment coefficients. McDonnell Douglas Corporation (MDC) datum obtained from wind tunnel tests (C_L,C_M see Reference [8], C_D see Reference [9]), have also been presented for comparison. As the VSAERO model did not include the LEX wake, the angle of attack was limited to $\alpha \leq 10°$ for realistic calculation of attached flow. Above this angle, leading edge separation and subsequent vortical flow, have a significant influence on the aerodynamic loading of the aircraft (Reference [10]). It can be seen that the VSAERO and the MDC results for lift coefficient compare reasonably well for both Mach 0.2 and Mach 0.6. At lower angles of attack VSAERO tends to over-estimate the lift slightly.

The VSAERO result for drag, presented in figure 7, shows trends similar to the MDC data, however the magnitude of the VSAERO results is much smaller. This is to be expected as the VSAERO predictions do not include corrections for viscous effects (i.e. only inviscid result obtained) as the option for viscous corrections was considered too expensive in computational time for the expected improvement in results. It is worth noting that the MDC drag coefficient data reproduced here are identical for both Mach 0.2 and Mach 0.6 over the range $\alpha = -10°$ to $+16°$.

Pitching moment results are also presented in figure 7. The predictions compare well with MDC data over a limited range, from $\alpha = 0°$ to $4°$. At $\alpha = 8°$ the VSAERO

result diverges markedly from the MDC data, indicating that a problem exists with the VSAERO model for this angle of attack. This problem may be a consequence of the lack of a LEX wake, or wake interactions. The absence of the LEX wake means vortex lift was not generated from the LEX. This would have affected the pitching moment substantially and also the overall lift coefficient.

To obtain further insight into the accuracy of the symmetric F/A-18 VSAERO model predictions, pressure distributions through various sections were compared with results from tests on the AOD 1/9th scale F/A-18 low speed wind tunnel model (Reference [5]). Results from the symmetric F/A-18 VSAERO model were obtained for $\alpha = 0°, 5°, 10°$, with a stabilator deflection of $+1°$, at Mach 0.15. These conditions matched particular cases presented in Reference [5]. These calculated pressure distributions are presented in figures 8, 9, 10, 11, and 12.

Figure 8 shows the pressure coefficient variation on a section through the aft fuselage at station y = 22 in. At $\alpha = 0°$ the VSAERO results for the upper surface compare well with the wind tunnel results of Reference [5]. As the angle of attack increases the magnitudes of the VSAERO predictions are less negative than the wind tunnel results, although similiar trends are still evident. On the lower surface, the wind tunnel results remain close to zero over the range of angle of attack considered here. At the x = 670 in station, VSAERO predicts pressure coefficients that are slightly more negative, and which change slope more abruptly than the wind tunnel results.

Figures 9, 10, and 11 show the pressure distributions over the stabilator for three angles of attack at stations y = 55 in (root), y = 93 in (mid-span) and y = 117 in (tip) respectively. In general the magnitudes and shapes of the VSAERO estimates and the wind tunnel (Reference [5]) pressure distributions are similiar. An interesting difference exists on the upper surface for $\alpha = 5°$ and $\alpha = 10°$ at the y = 93 in (mid-span) and y =117 in (tip) stations. The leading edge suction peak predicted by VSAERO follows the classical two dimensional pressure distribution, while the wind tunnel results indicate a plateau region at the leading edge. This pressure distribution is indicative of a vortex flow associated with a swept wing with a sharp leading edge. Reference [11] describes the formation and behaviour of the wake which separates from the leading edge of a swept wing as a function of angle of attack. For the lower surface the pressure distributions compare well.

The VSAERO and wind tunnel results (Reference [5]) plotted in figure 12 show the effect of angle of attack on the fin pressure distribution at station z = 172 in. The figure indicates differences between the VSAERO and the wind tunnel pressure distributions on the inner surface for the cases of $\alpha = 5°$ and $\alpha = 10°$. The VSAERO result shows only the pressure distribution associated with attached flow and does not include the effects of flow changes due to the LEX vortex or the vortex separating from the leading edge of the fin. On the outer surface (positive pressure) the VSAERO and wind tunnel results compare well.

The results presented show that the symmetric F/A-18 VSAERO model would not be able to provide accurate pressure distributions for angles of attack and stabilator deflections where separated vortex flow becomes significant. Previous attempts to model separated vortex flow from the LEX of this F/A-18 VSAERO model had been unsuccessful at AOD. Hence further development of the symmetric F/A-18 model to include the LEX vortex and vortex separations on the stabilator and fin was not undertaken for this task.

4.2 Roll Results

Although the VSAERO program is capable of calculating aerodynamic loadings for steady rotations (i.e. pitch, roll or yaw rates) the problem of wake interference discussed previously makes the procedure very difficult when modelling roll rates of complex aircraft configurations. To undertake this work would have required development of an asymmetric F/A-18 model, utilizing approximately 5000 panels, from the existing half model geometry (i.e. the symmetric F/A-18 model). Computational time for such a large VSAERO model was considered excessive, and was not pursued. Therefore, an investigation was carried out to determine if the starboard stabilator and fin model could be used in isolation to calculate load distributions for steady roll conditions. It was planned to determine loading increments using the VSAERO steady rotation capability for $\alpha = 0°$, $\beta = 0°$, with Mach 0.5 at sea level for a maximum roll rate of $p_{x'} = -3$ rad/s. The direction of rotation is defined in figure 3. The maximum angle of attack due to the roll rate occurs at the tip sections of the fin and stabilator. Within these limiting conditions it was assumed that the results could be scaled to the particular tip section angle of attack experienced during a manoeuvre and then added to the load distribution calculated for the basic static angle of attack, using the symmetric F/A-18 VSAERO model.

However this approach proved to be unsatisfactory since, whenever vortex flow conditions exist, superposition of two linear solutions is inaccurate when modelling the non-linear vortex lift which is characteristic of leading edge vortex separation. References [12] and [13] give further information on a suitable method for modelling this type of flow. A strip theory method was therefore used to calculate the incremental effect of steady roll rate. While the strip theory method provides only a first order approximation to the aerodynamic loading during steady rotation, it is considered to be more accurate than using the VSAERO model when separated vortex flows and wing wake interference effects are present.

Figure 13 presents the steps involved in obtaining the incremental loading using strip theory. Figure 13a shows the basic wind tunnel data (Reference [5]) for the stabilator at deflection angles of $+1°$ and $+9°$. It indicates the presence of vortex lift with the peak load at $Y_H/b_{stab} = 0.58$. From figure 13a, the load increment per degree of stabilator deflection is obtained using linear interpolation (see fig. 13b). Next the angle of attack is calculated based on the stabilator y coordinate, roll rate and velocity (see fig. 13c). Finally figure 13d shows the incremental load on the starboard stabilator due solely to the roll rate. Obviously the leading edge vortex has had a major influence on the load distribution, forcing the peak load outboard.

Results obtained using the strip theory method must be considered with some caution. If the combined angle of incidence due to roll rate and initial angle of attack is less than the critical angle for the existence of vortex flow, then using the aerodynamic data in Reference [5], which effectively scales a vortex flow, will be inaccurate. This method also assumes that the vortex flow due to the combined effects of angle of attack and rolling behaves in the same way as a vortex flow due to a steady angle of attack.

Figure 14 shows the spanwise load distribution due to a roll manoeuvre obtained from the VSAERO stabilator and fin model, and also the result from the strip theory method described previously. The parameters used for the VSAERO calculation were the same as those used for the strip theory method above. The result from the

strip theory method is substantially greater than the result obtained from VSAERO. Also the peak loading is at $Y_H/b_{stab} = 0.58$ for strip theory, whereas VSAERO predicts the peak loading further outboard at $Y_H/b_{stab} = 0.73$. A zero load condition was imposed at the tip for the strip theory method, while the VSAERO result does not include the results from the tip patch where the loading would return to zero.

4.3 Sideslip Results

Two VSAERO models were used to obtain results for the F/A-18 fin with rudder deflected, as this information was not available from the wind tunnel tests (Reference [5]). The symmetric F/A-18 model was used to obtain results for zero sideslip with rudder deflection, while the starboard fin model was used for cases involving combinations of angle of attack, sideslip and rudder deflection. Rudder deflections of $\delta_r = 0°, \pm15°, \pm30°$ were covered with both models. Sideslip angles of $\beta = 0°, \pm10°$, and $\alpha = 0°, 10°$ were covered using the starboard fin model, while the symmetric F/A-18 model used only $\alpha = 0°$, $\beta = 0°$. To obtain a complete set of results using the starboard fin model both positive and negative sideslip angles were computed. In all cases Mach 0.25 was used, and the option to include viscous corrections was not utilized for the reasons given previously in Section 4.1. It is also worth noting that the symmetric F/A-18 model (half model), with starboard rudder deflected, effectively models either both rudders deflected inboard, or both deflected outboard. This limitation of the symmetric F/A-18 model is unlikely to have a important effect on the results obtained because of the lateral separation of the fins.

Results from the starboard fin model described in Section 3.3 were compared with results from two sources. For the case of zero sideslip with rudder deflection, the results from the symmetric F/A-18 model were used. For the cases of non-zero sideslip with no rudder deflection, results from wind tunnel tests (Reference [5]) were used.

Figure 15 presents graphs of fin spanwise load distributions for the starboard fin from the symmetric F/A-18 model, and also from the starboard fin model. The results from the symmetric F/A-18 model with zero rudder deflection show the loading is slightly negative at the root and changes sign towards the fin tip. With rudder deflected the maximum loading occurs approximately at the mid-span of the rudder, and is seen to be approximately symmetric for both positive and negative rudder deflections. The increase in loading on the fin caused by the rudder deflection also carries over to the tip section of the fin. The spanwise load distribution does not display any extreme variations at the root section, or at the rudder tip - outer fin junction (i.e. $Z_V/b_{fin} = 0.7$). The starboard fin model results and the symmetric F/A-18 model results show a number of minor differences. Firstly, the magnitude of the starboard fin model load distribution is slightly less than the symmetric F/A-18 fin results. Also the starboard fin model results have greater variations at the root section and also at the rudder tip - outer fin junction. These variations are probably due to the effects of the wakes and the local flow about the intersection of the patches. Presumably the higher panel density of the starboard fin model has been able to predict these local flow effects more accurately than the symmetric F/A-18 model. Finally, for the zero rudder deflection case, the fin setting angle with respect to the fuselage centreline, or 'toe-out' angle, of 1° has produced a slight

positive offset of the spanwise loading for the starboard fin model. This effect is not observed in the result from the symmetric F/A-18 model, indicating that the wing and fuselage have an effect on the fin spanwise lift distribution.

Figure 16 shows the effect of angle of attack on the spanwise lift distributions predicted using the starboard fin model. It can be seen that the fin cant angle of 20°, combined with $\alpha = 10°$, has caused all of the spanwise load distributions to be displaced in a negative direction (i.e. inboard). For the zero rudder deflection case, the value of the fin spanwise local lift coefficient, C_{lf}, is approximately -0.1. Large variations are again present at the edges of each patch, i.e. the tip and root sections, and also at the rudder tip - outer fin junction. If these regions are ignored, and the additional load increment due to each rudder deflection considered, it can be seen that the additional load increments are of similar magnitude at each angle of attack.

Figures 17 and 18 show the effect of sideslip angle on the fin spanwise loading at $\alpha = 0°$ and $\alpha = 10°$ respectively. A negative sideslip is seen to displace the spanwise load by a positive value (i.e. outboard), and vice versa. Increasing the angle of attack again offsets the spanwise loading, but this is a relatively minor change compared with the effect due to sideslip angle.

Integrated results for fin normal force, bending moment and pitching moment coefficients from both VSAERO and wind tunnel tests (Reference [5]) are presented in figure 19. VSAERO results cover $\alpha = 0°, 10°$, with $\beta = 0°, \pm 10°$ and rudder deflections $\delta_r = 0°, \pm 15°, \pm 30°$. The wind tunnel results cover $\alpha = 0°, 10°$, with $\beta = 0°, \pm 5°, \pm 10°$ and a rudder deflection of $\delta_r = 0°$. For the $\alpha = 0°$, $\delta_r = 0°$ case, the results are comparable, as the gradients match over the $\beta = -5°$ to $5°$ range. The slight increase in gradient for the wind tunnel results when the sideslip angle exceeds $\pm 5°$ is due to the leading edge vortex on the fin. For the $\alpha = 10°$ case the wind tunnel results differ considerably from the VSAERO results. This is probably due to the LEX vortex beginning to affect the flow and modifying the fin lift at this angle of attack.

5 Conclusions

VSAERO has been used to estimate aerodynamic load distributions on the aft fuselage, fins and stabilators of the F/A-18 aircraft for a number of test conditions. Due to the complex aerodynamics associated with vortex flows on the aircraft, VSAERO has not been capable of providing accurate results for the full range of conditions initially considered. Specifically, VSAERO cannot adequately model the leading edge separations and the vortex flows that are present on thin aerofoil sections such as the stabilators and fins at moderate to high local incidence angles. Difficulties also arose with a number of VSAERO solutions when the wing wakes intersected with the fuselage and stabilator panels, causing erroneous pressures to be predicted. Wake interference problems were avoided by constraining the wake, so that surfaces could not be intersected.

Comparisons of the VSAERO predictions with the wind tunnel results (Reference [5]) have been made for a number of test conditions. The comparisons provide information on the level of accuracy and the extent to which this type of panel code can be applied to modern fighter type configurations.

Simplified models of the starboard fin, and also the starboard stabilator and fin, were used to obtain results for cases with sideslip and rudder deflection, and roll conditions. Results from the simplified models were compared with the results from the symmetric F/A-18 model for zero sideslip cases, and were then used to provide information on the incremental load distributions due to sideslip angle and rudder deflection. The results for a roll manoeuvre from the starboard stabilator and fin model were compared with strip theory calculations using data from Reference [5].

Overall, the VSAERO predictions provided valuable initial information on expected pressure distributions and for establishing wind-tunnel test requirements and data handling and data presentation formats for this task.

Acknowledgements

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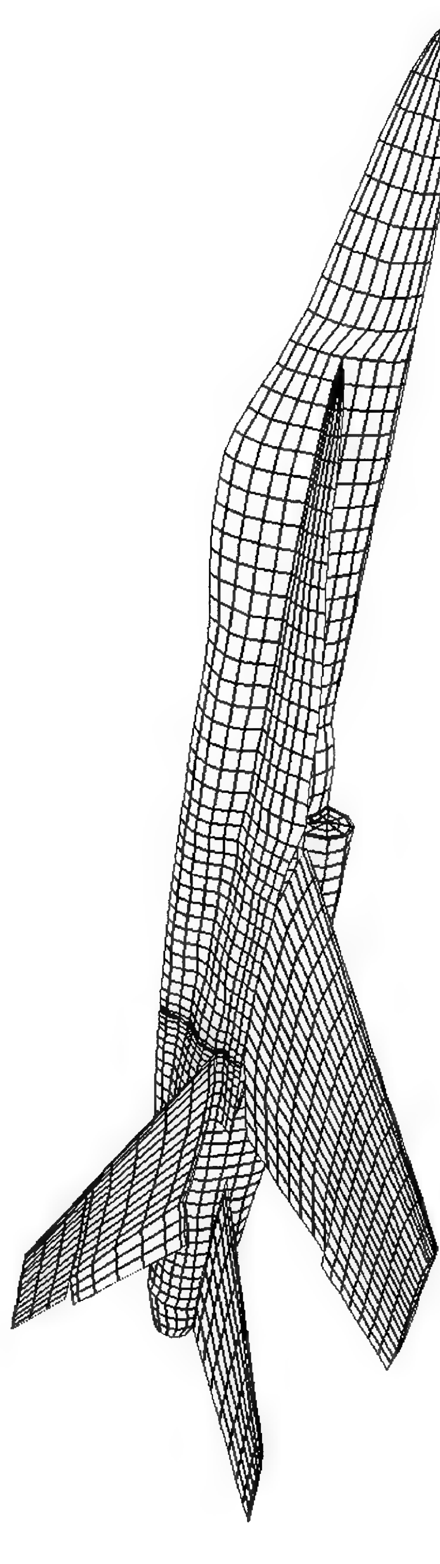


Figure 1: Symmetric F/A-18 VSAERO Model

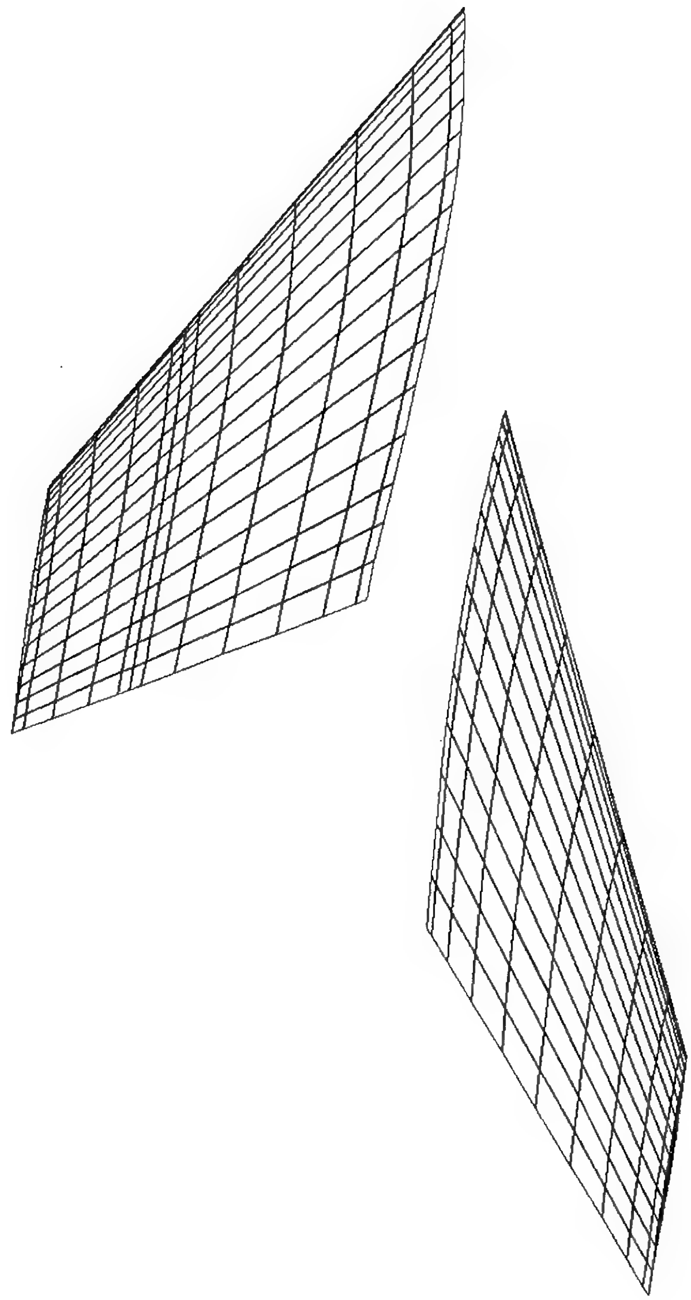


Figure 2: F/A-18 Starboard Stabilator and Fin VSAERO Model

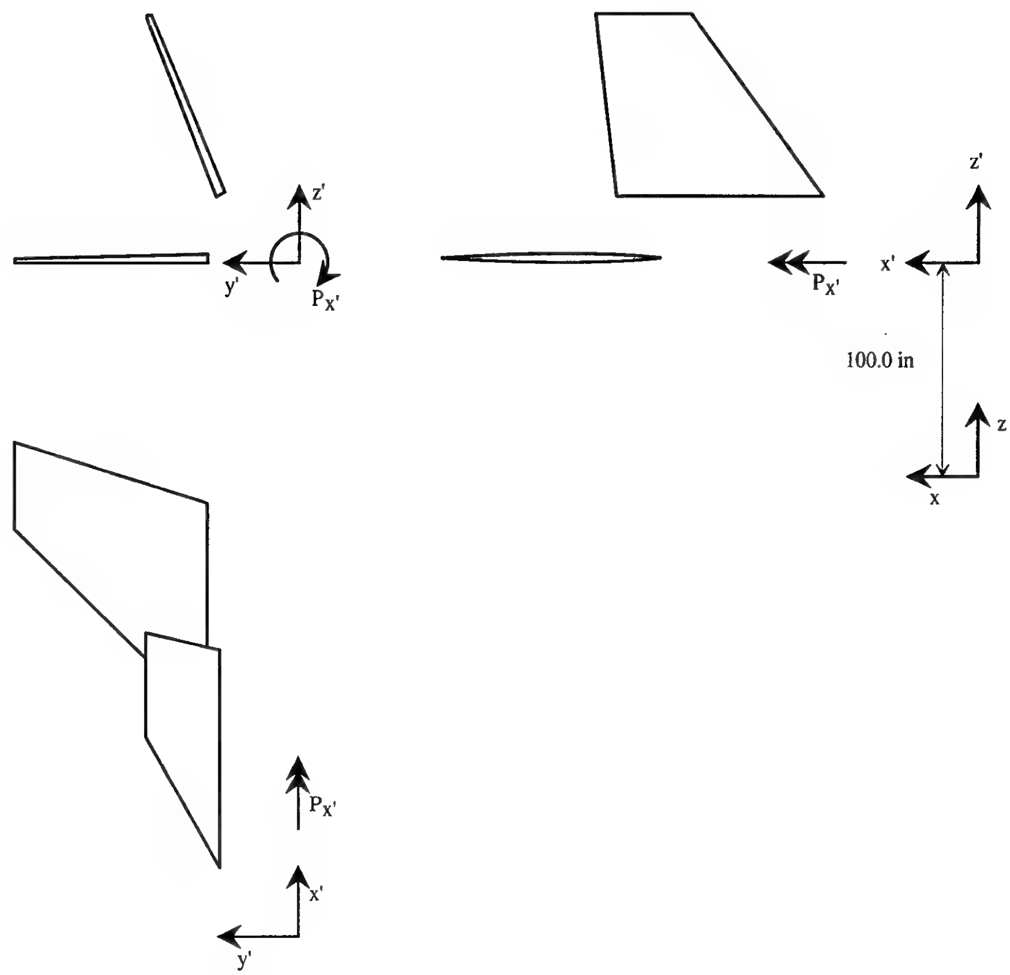


Figure 3: Positive Sign Convention for Roll Rotation on the Starboard Stabilator and Fin VSAERO Model

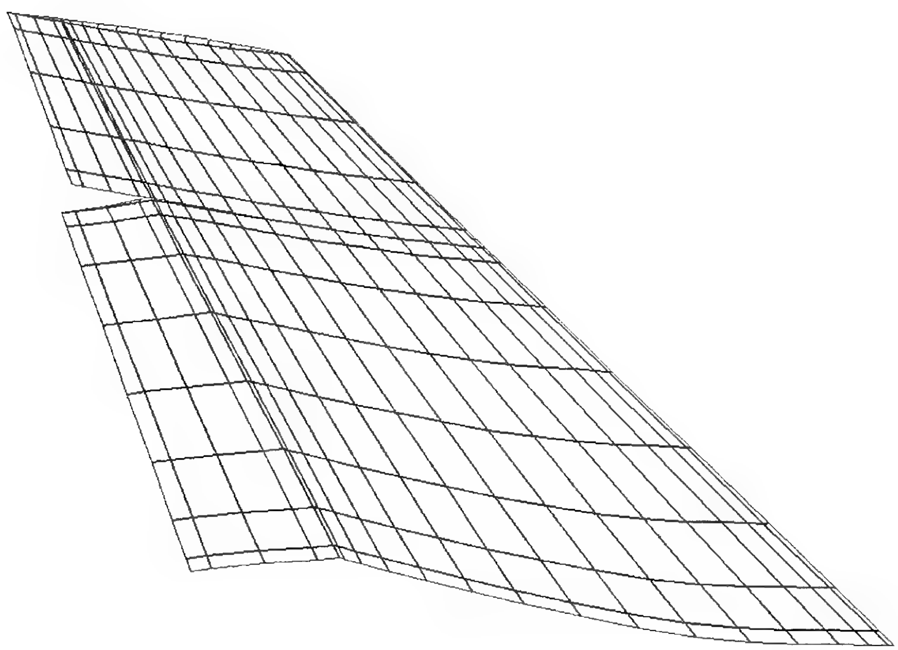


Figure 4: F/A-18 Starboard Fin VSAERO Model ($\delta_r = +30°$)

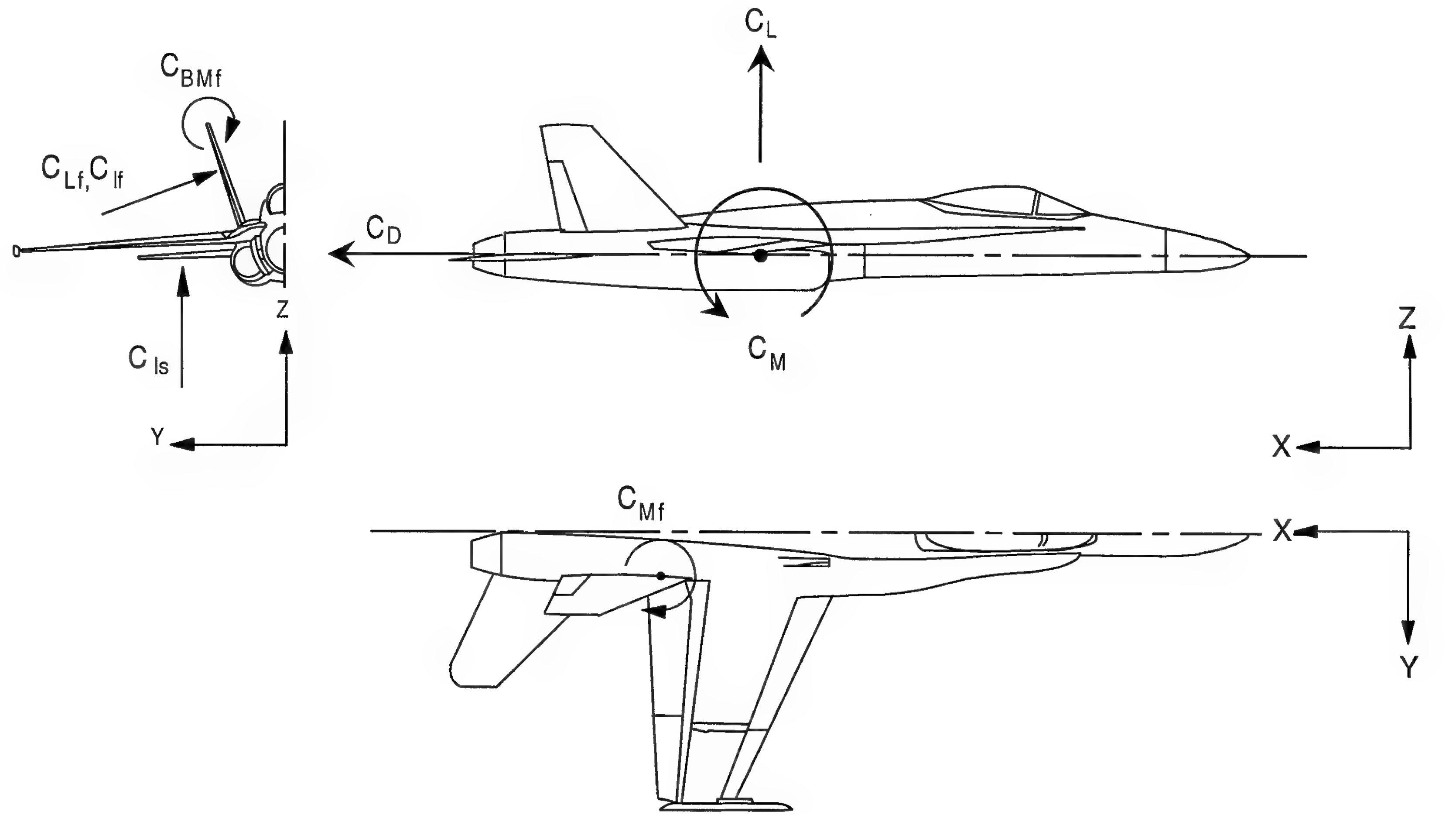


Figure 5 : Positive Sign Convention for Loads on the F/A-18 Aircraft

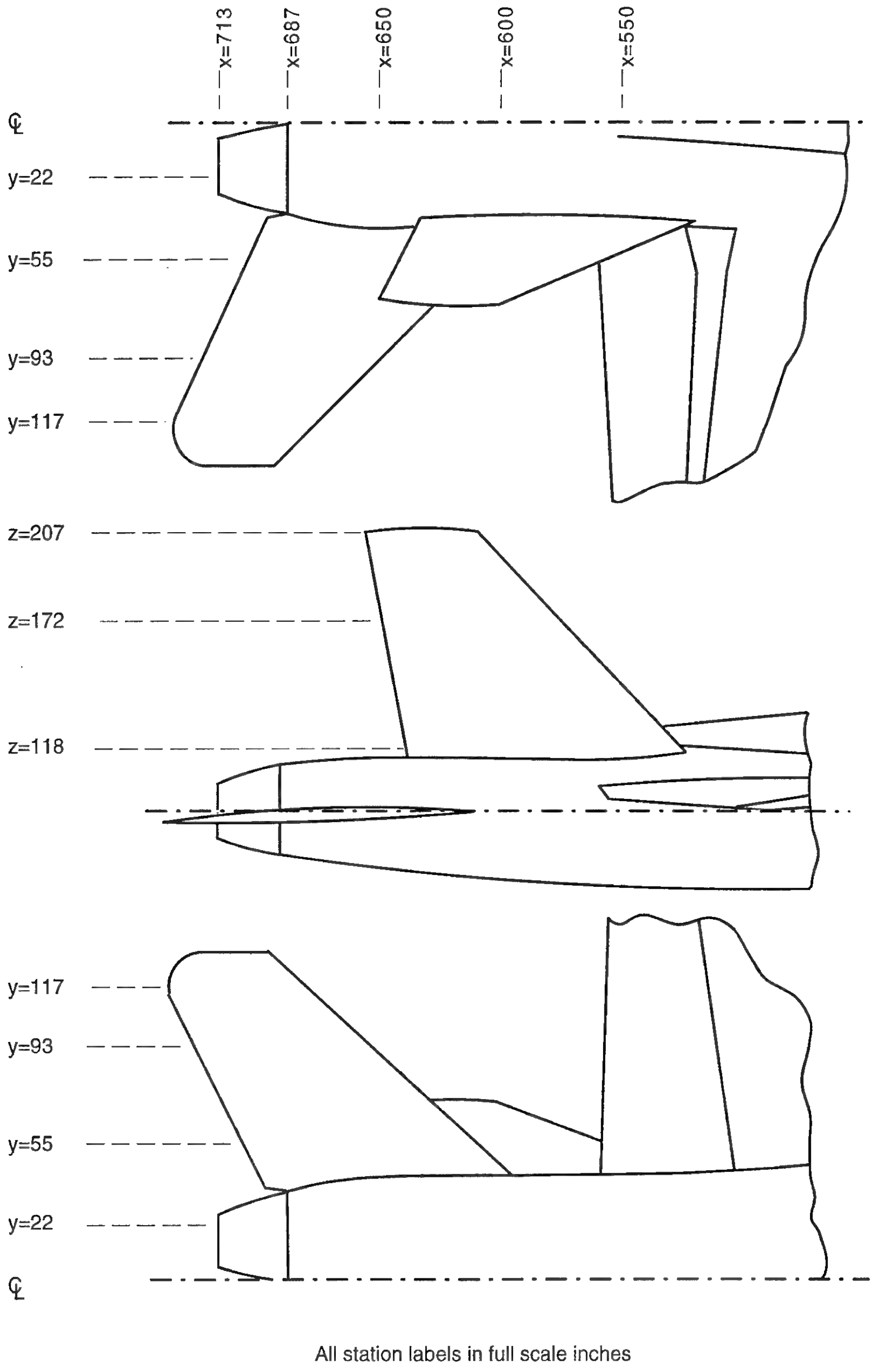


Figure 6: Reference Stations on Aft-Fuselage, Fin and Stabilator

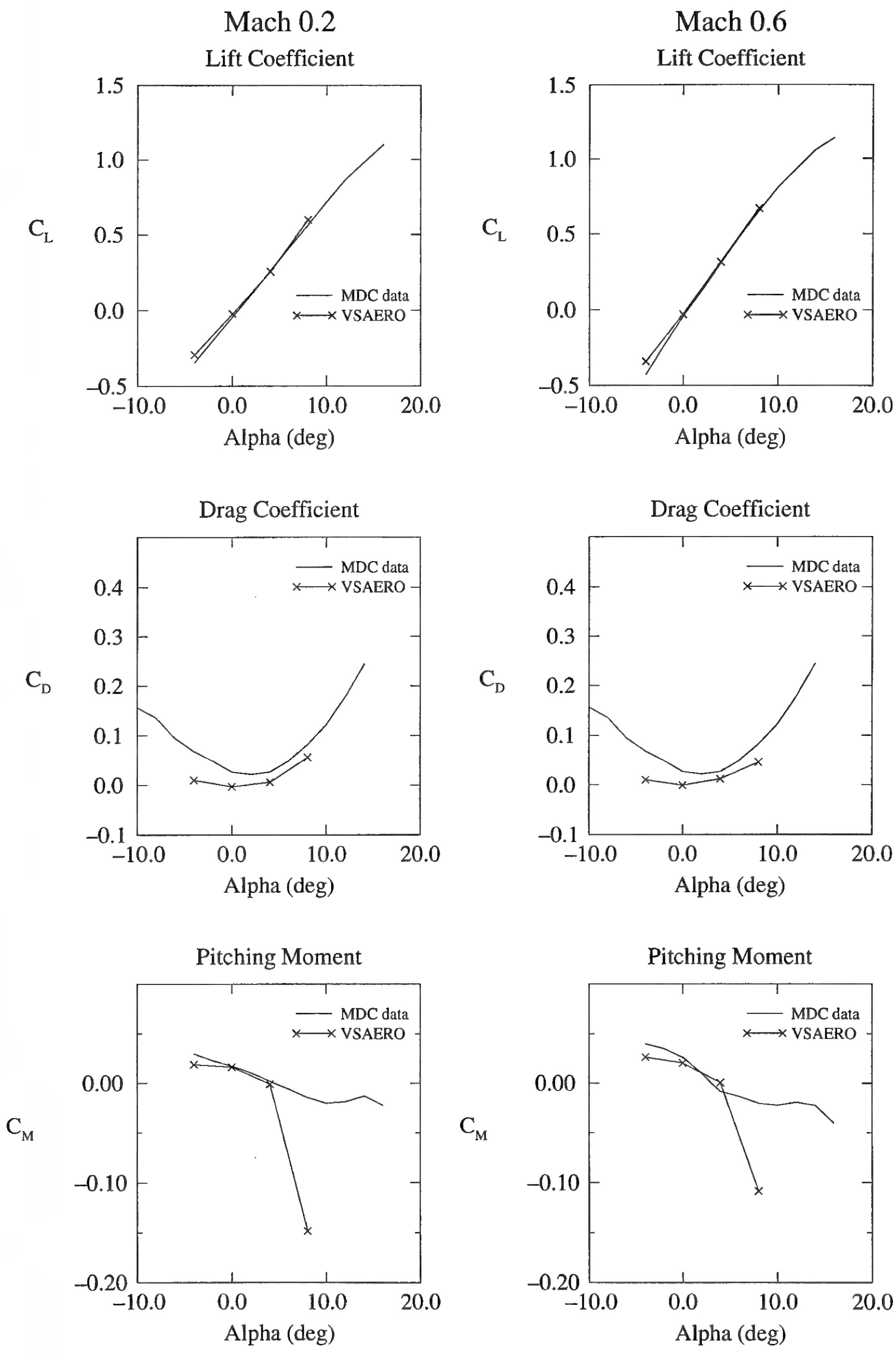


Figure 7: F/A-18 Lift, Drag and Pitching Moment Coefficients

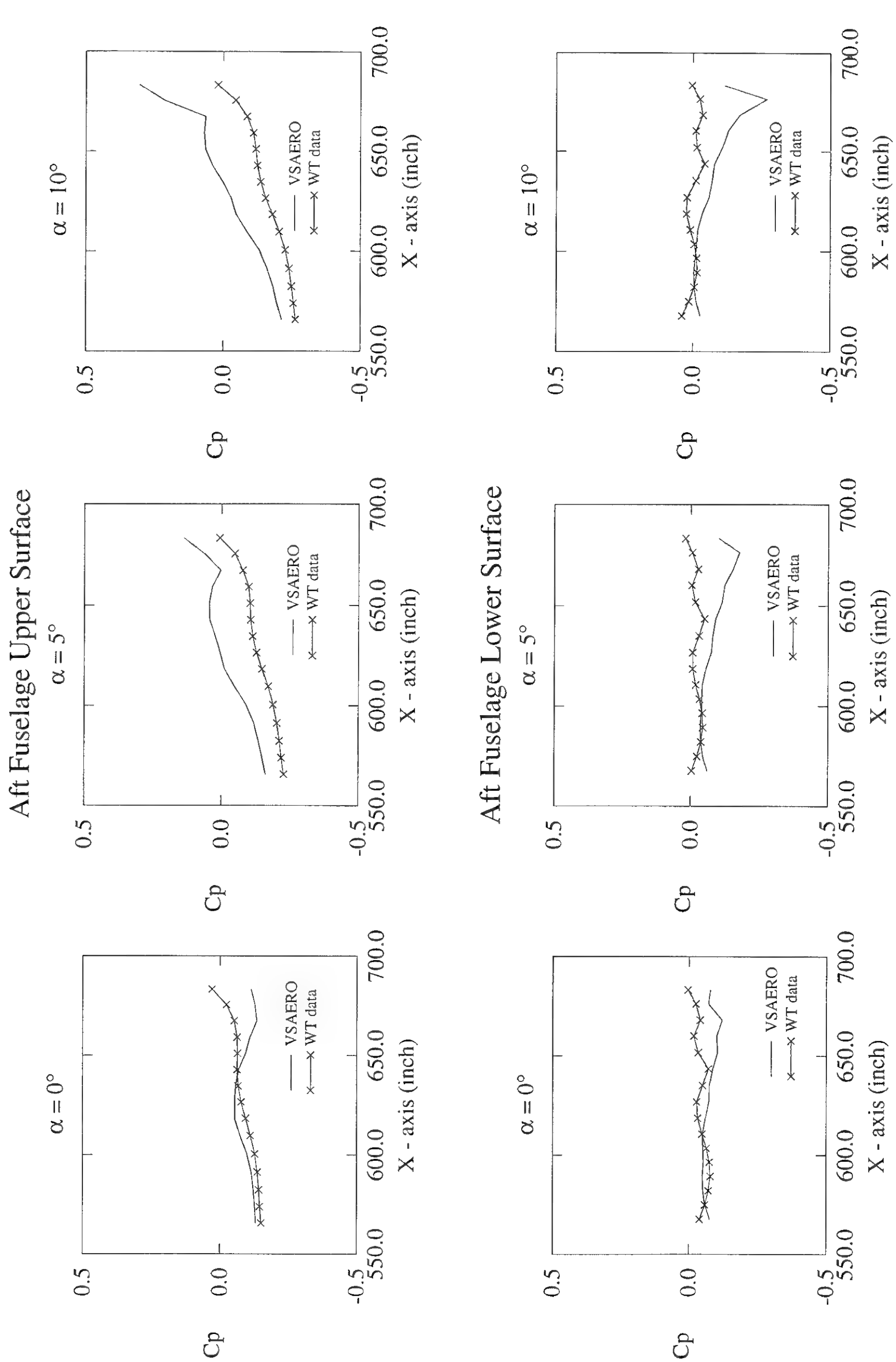


Figure 8: Fuselage Pressure Distribution (y station = 22 in, mid-fuselage)

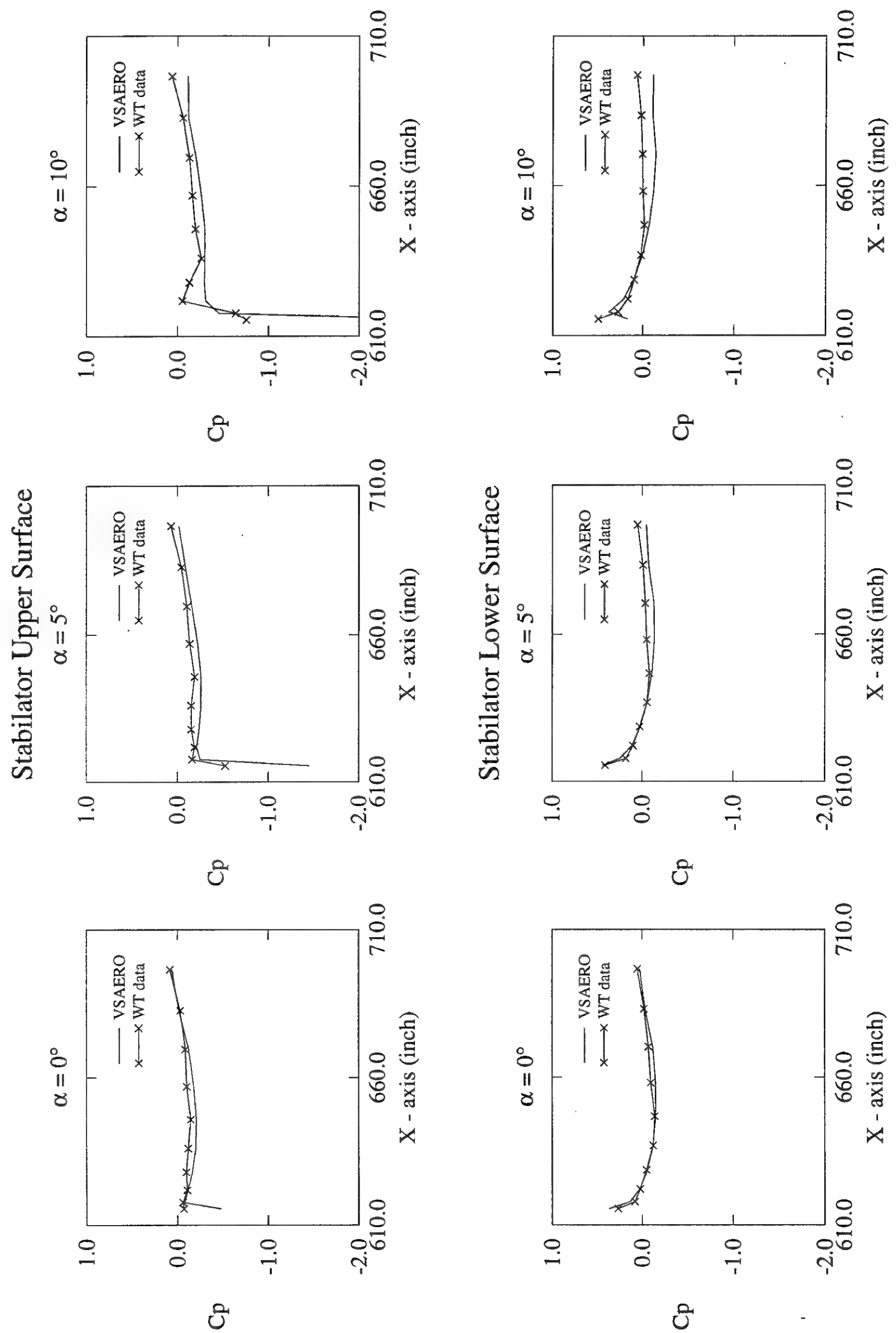


Figure 9: Stabilator Pressure Distribution (y station = 55 in, root section)

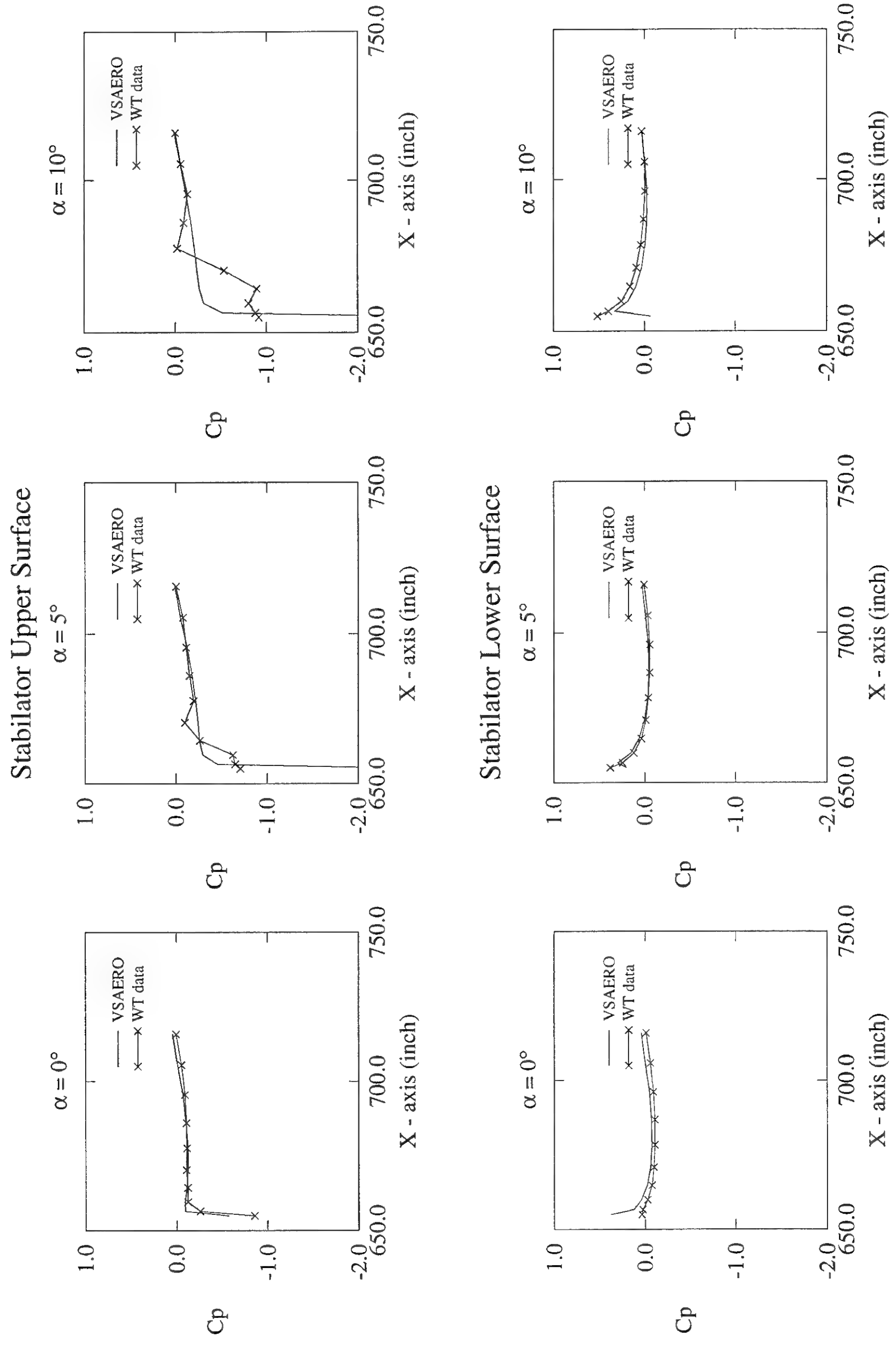


Figure 10: Stabilator Pressure Distribution (y station = 93 in, mid-span)

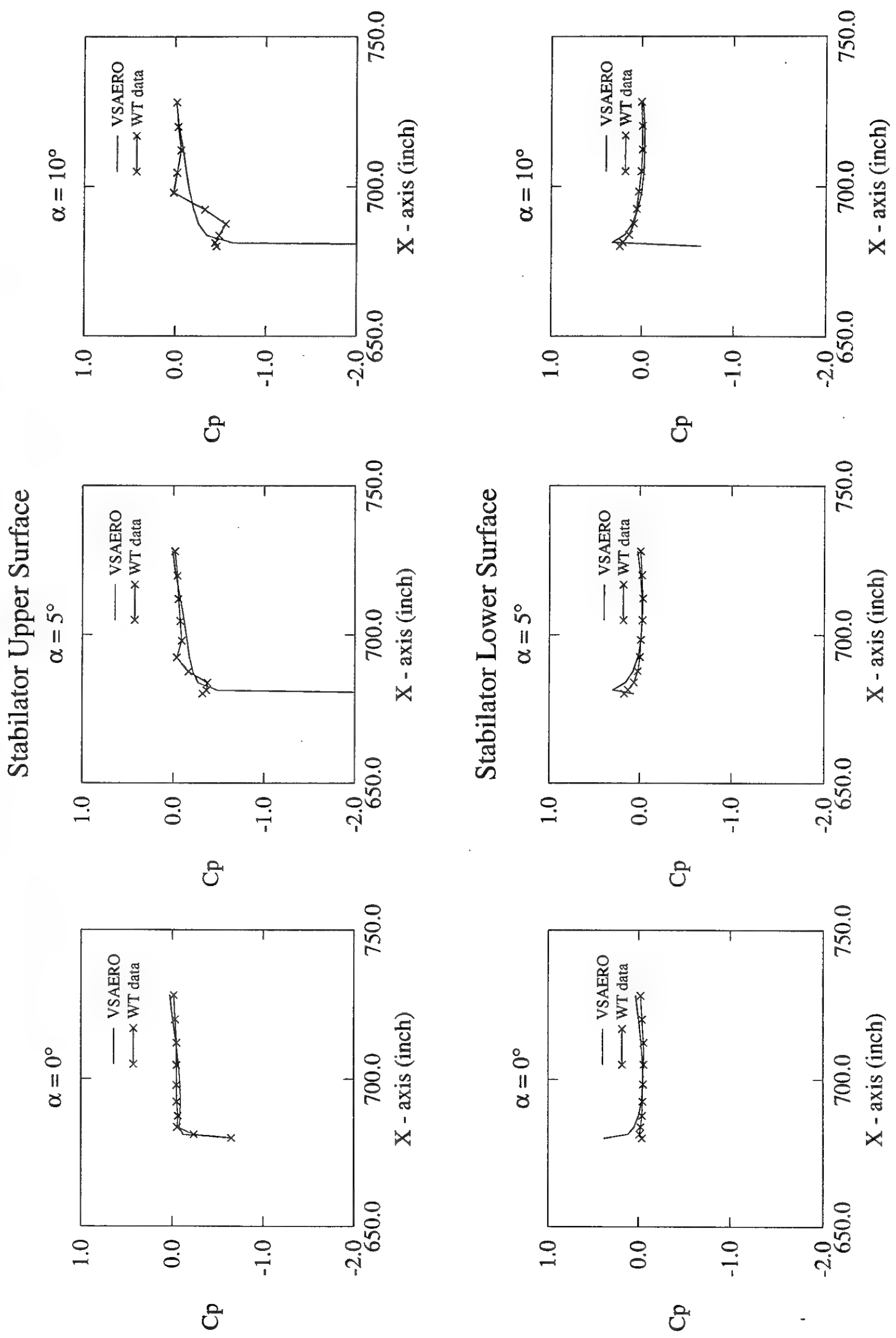


Figure 11: Stabilator Pressure Distribution (y station = 117 in, tip section)

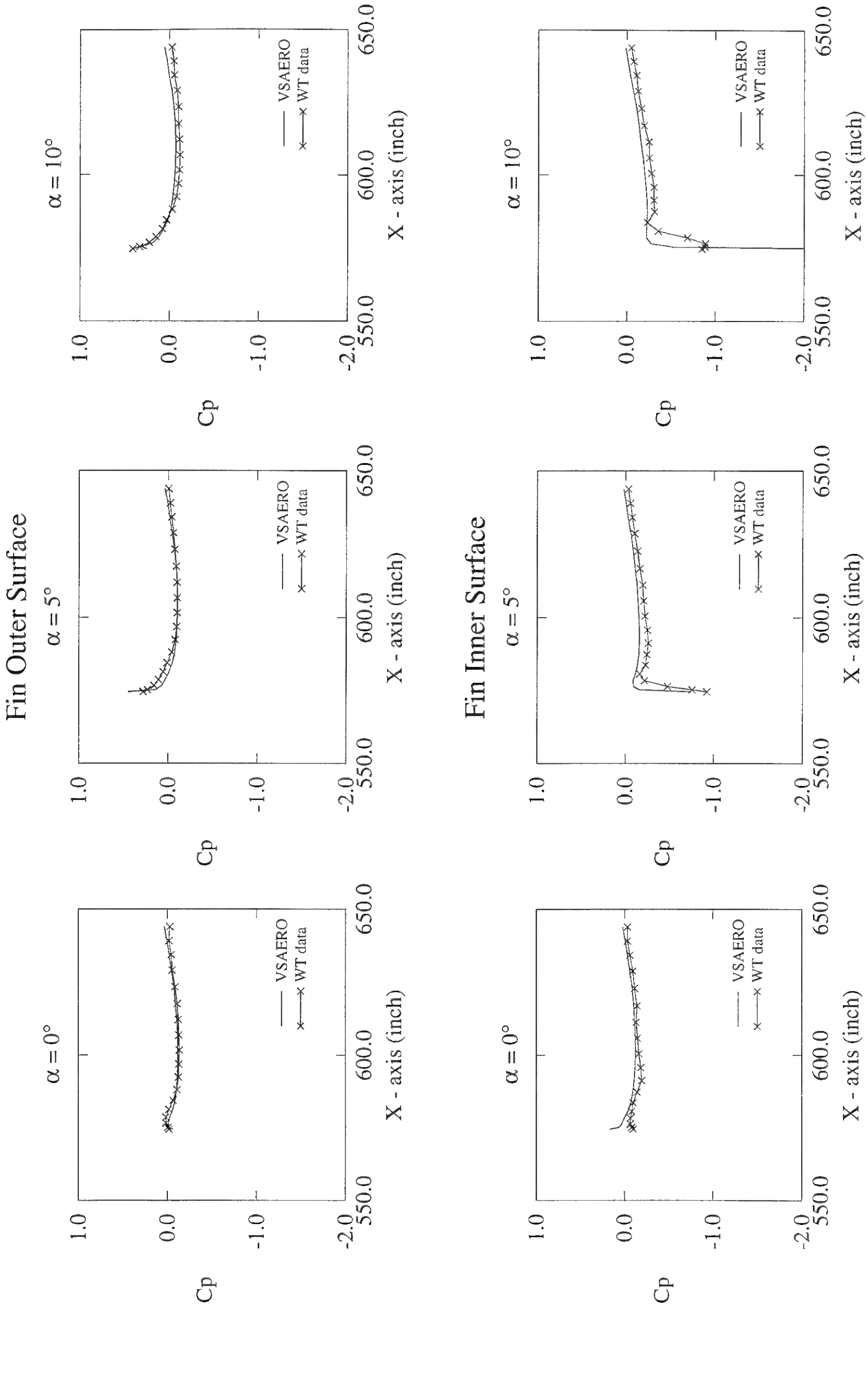


Figure 12: Fin Pressure Distribution (z station = 172 in, mid-span)

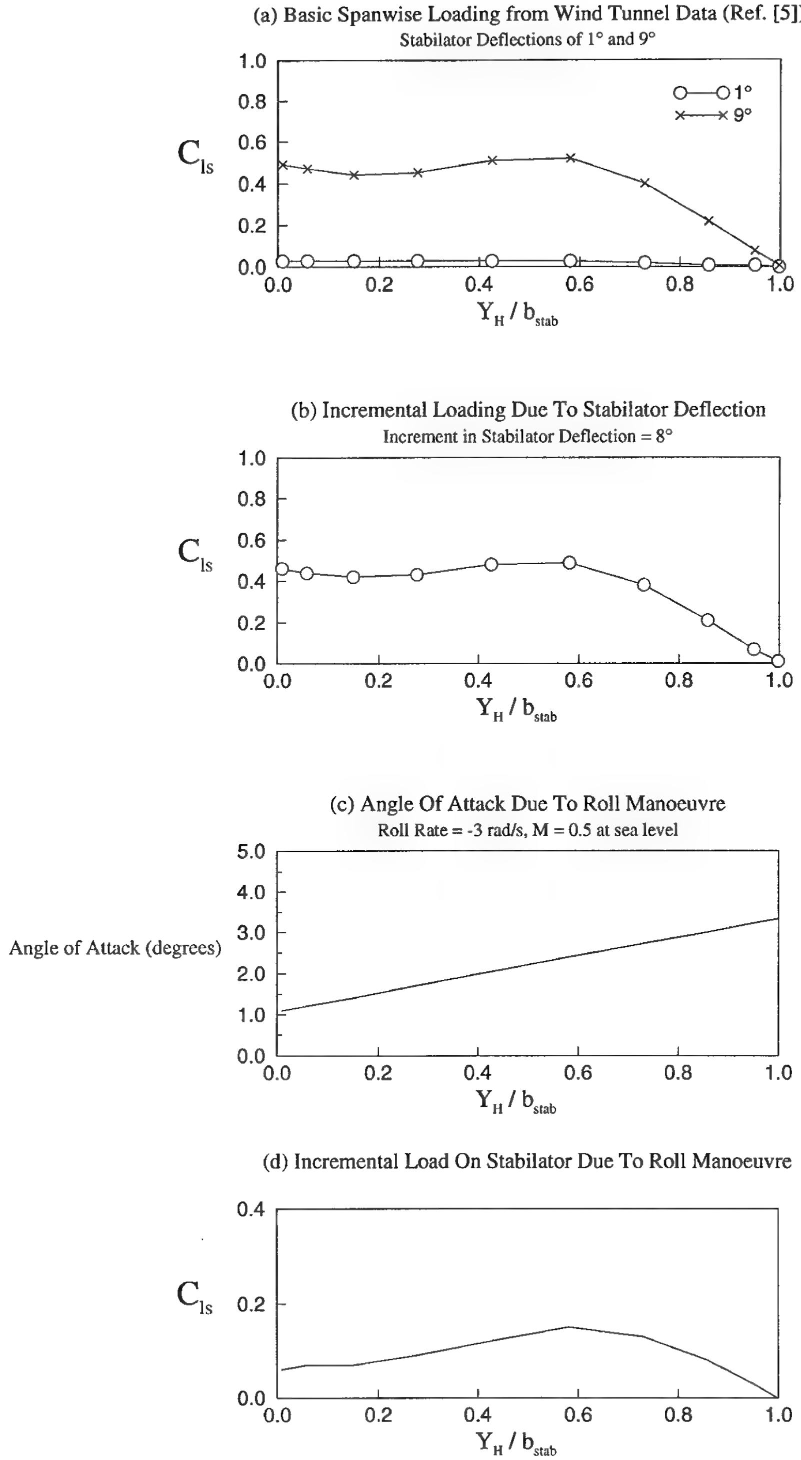


Figure 13: Incremental Loading on Stabilator due to Roll Manoeuvre (Strip Theory Method, Roll Rate = -3 rad/s, M = 0.5 at Sea Level)

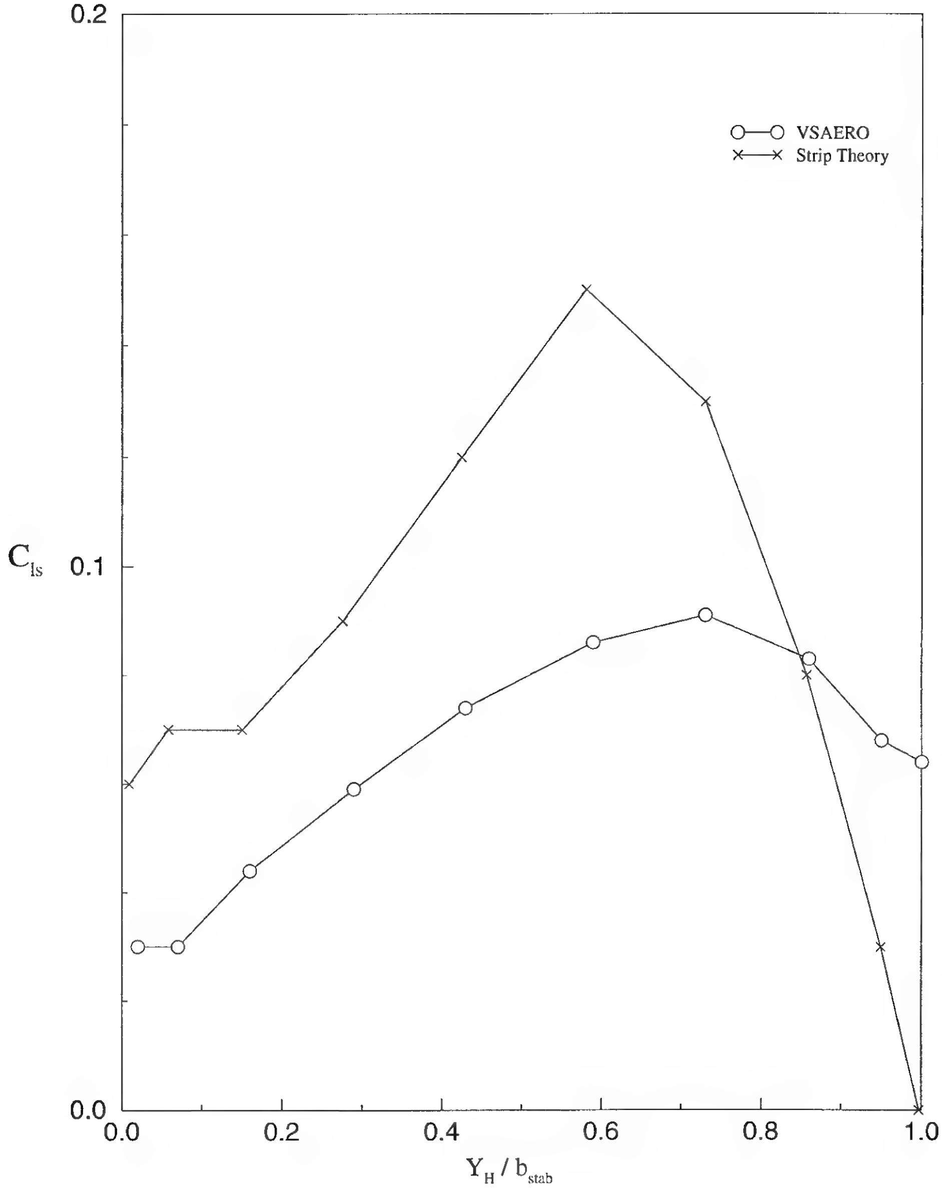
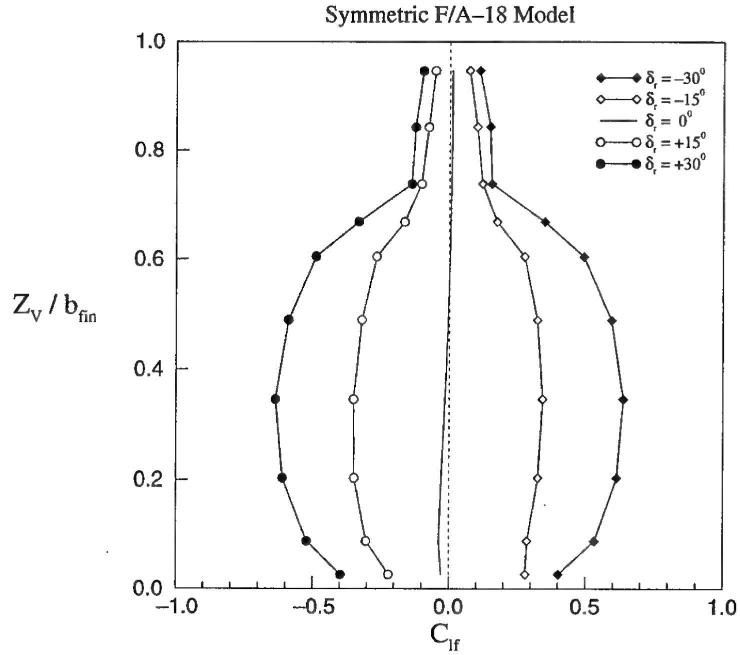


Figure 14: Comparison of Spanwise Load Distributions on the Stabilator due to a Roll Manoeuvre Calculated using Strip Theory Method and the VSAERO Starboard Stabilator and Fin Model



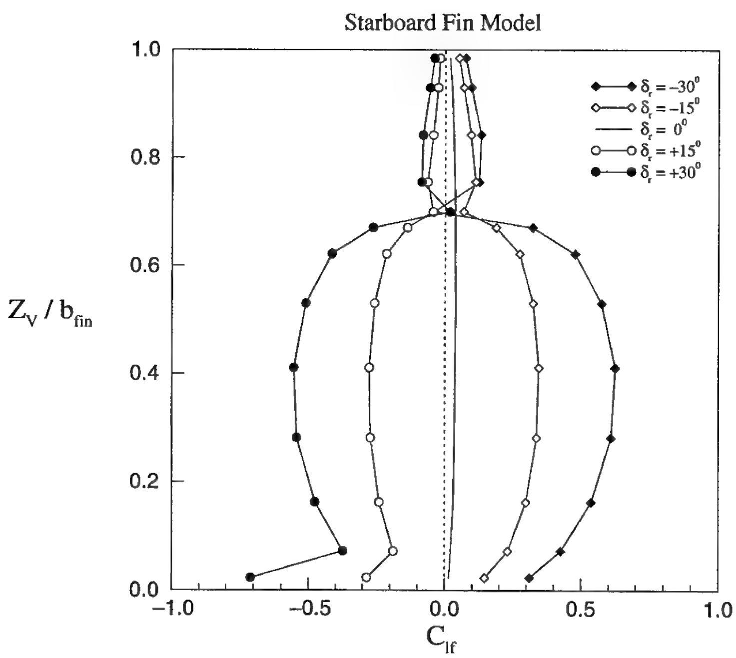
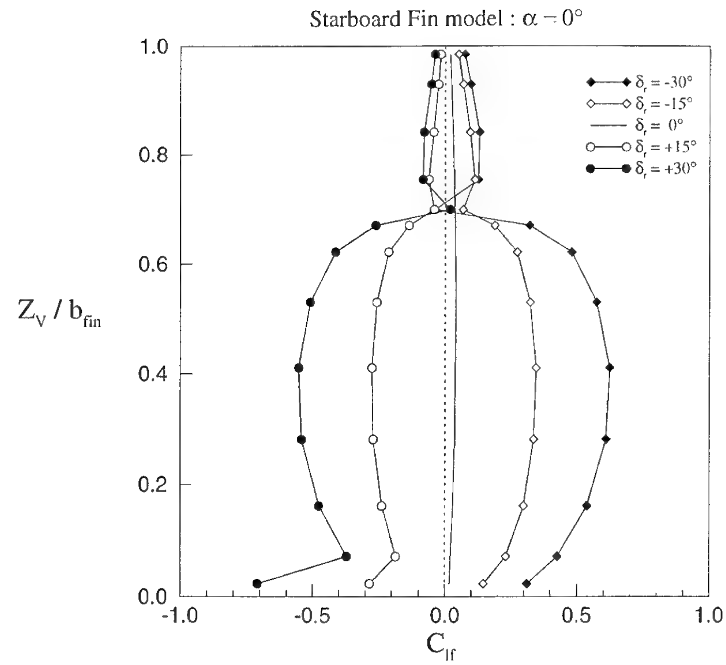


Figure 15: Fin Spanwise Load Distribution for Varying Rudder Deflection: Symmetric F/A-18 Model and Starboard Fin Model ($\alpha = 0°$, $\beta = 0°$)



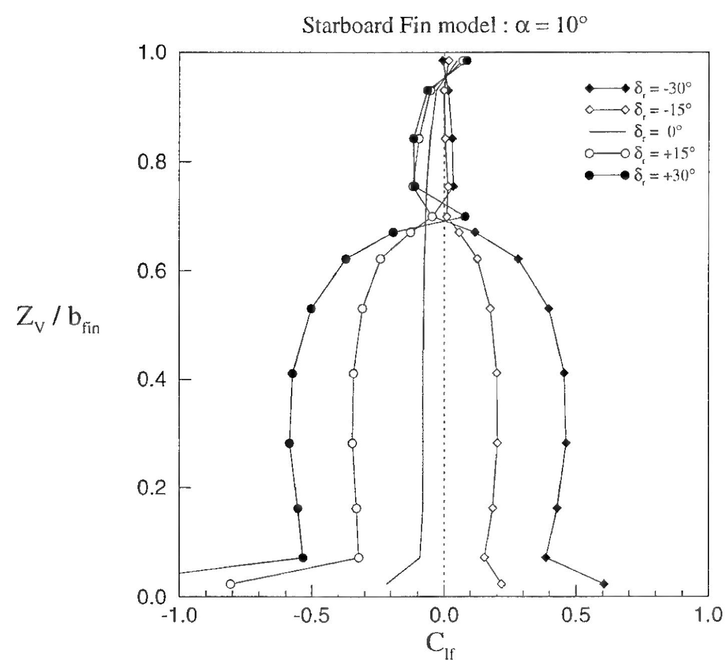
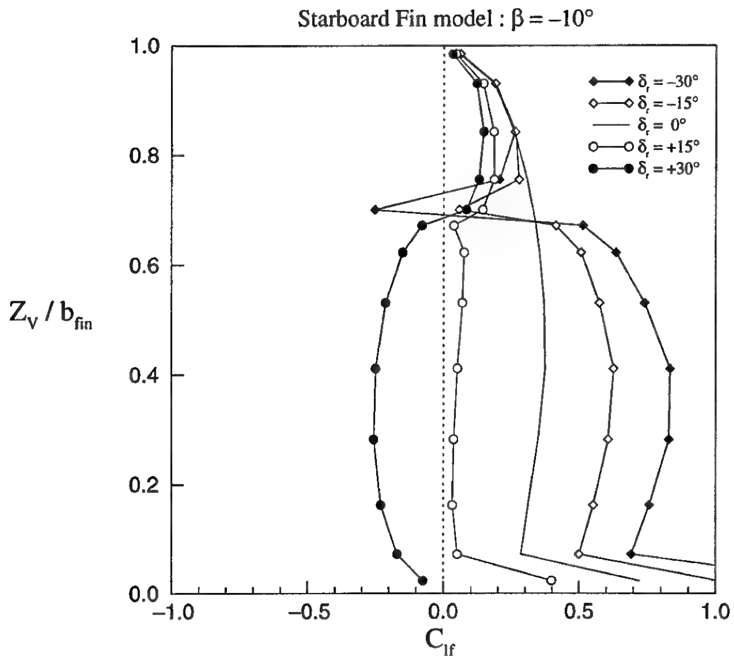


Figure 16: Fin Spanwise Load Distribution as a Function of α and δ_r : Starboard Fin Model ($\beta = 0°$)



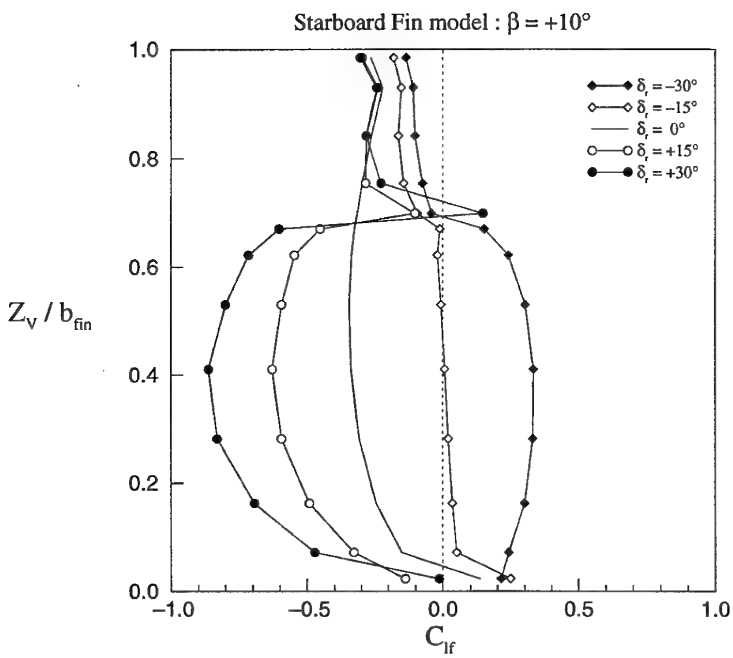
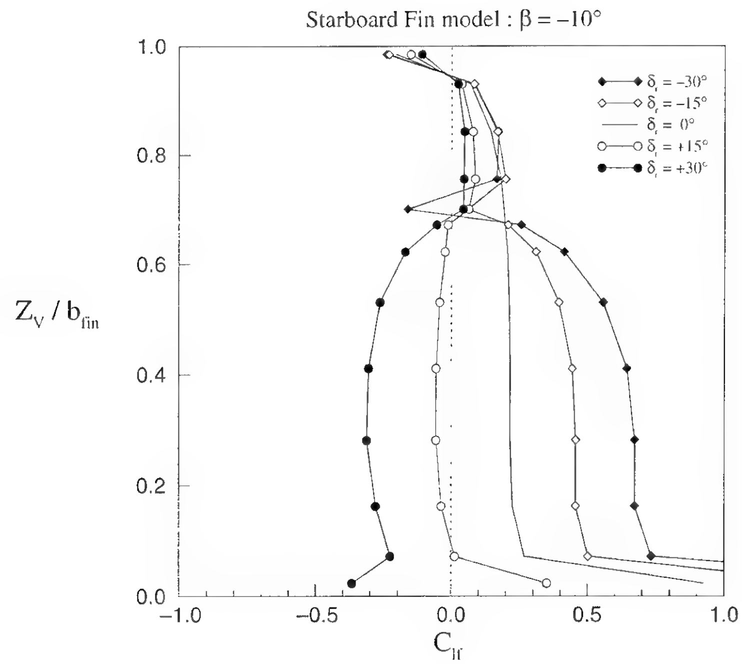


Figure 17: Fin Spanwise Load Distribution as a Function of β and δ_r : Starboard Fin Model ($\alpha = 0°$)



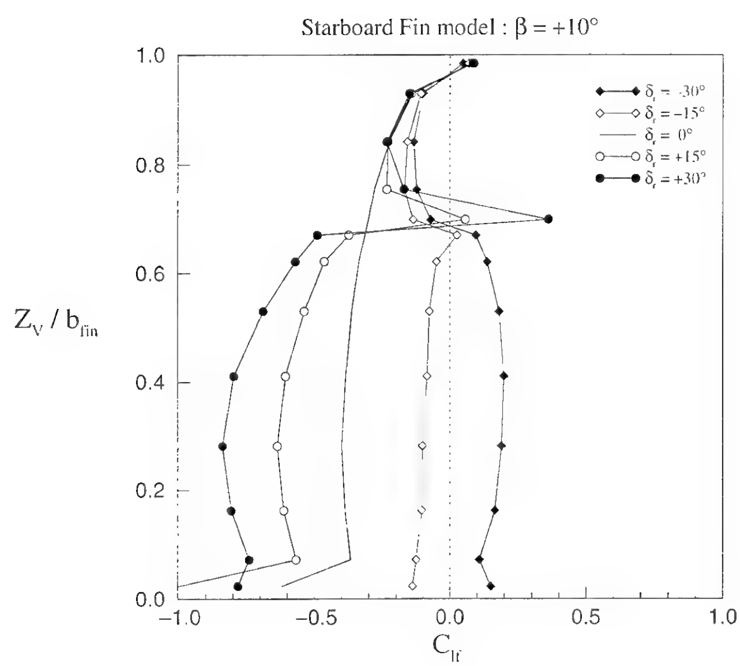


Figure 18: Fin Spanwise Load Distribution as a Function of β and δ_r : Starboard Fin Model (α = 10°)

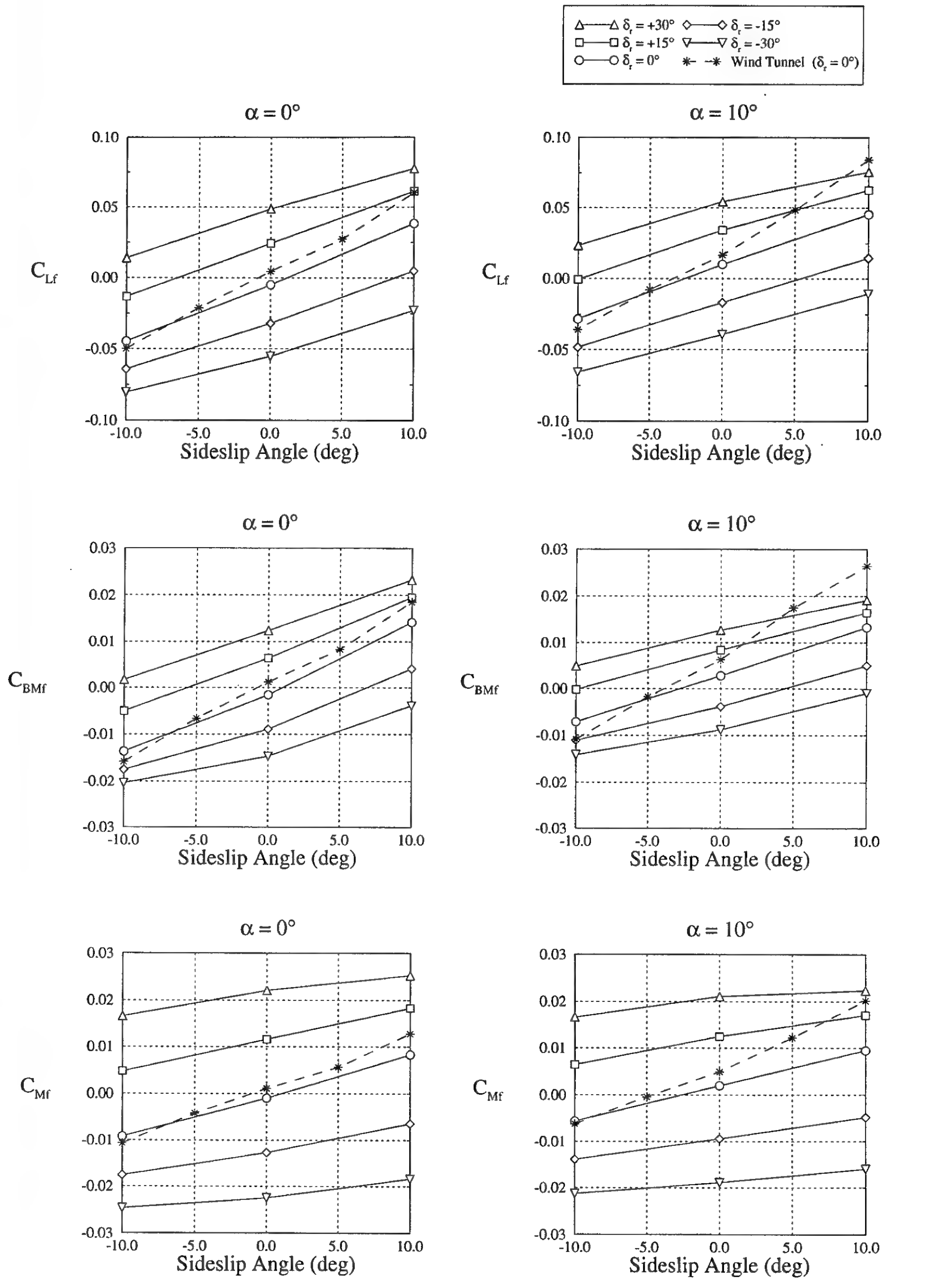


Figure 19: Normal Force, Bending Moment, and Pitching Moment as a Function of α, β and δ_r : Starboard Fin Model

Estimation of Aerodynamic Load Distributions on the F/A-18 Aircraft
Using a CFD Panel Code

H.A. Quick

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Fins	Rolling
Aft fuselages	VSAERO (computer program)

16. ABSTRACT

The computational fluid dynamics panel method code VSAERO has been used to estimate the aerodynamic load distributions on the stabilators, fins and aft fuselage of the F/A-18 aircraft for steady sideslip and steady roll conditions. Three separate VSAERO models have been developed to obtain results for these conditions. These are a symmetric F/A-18 model, a starboard fin model and a starboard stabilator and fin model. Difficulties in modelling the vortices separating from the leading edges of the stabilators and fins at moderate angles of attack have restricted the range of conditions for which the method has been used.

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